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Aerospace and Mechanical Engineering

Telephone
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Apr. [illegible], 1983

Mr. Dan M. Somers
NASA
Langley Research Center
Mail Stop 339
Hampton, VA 23665

Subject: Semi-Annual Status Report for the period November 1982 - April 1983 for NSG 1419, entitled, "The Structure of Separated Flow Regions Occurring Near the Leading Edge of Airfoils - Including Transition."

Dear Mr. Somers:

Enclosed you will find three copies of the subject report. Two additional copies have been sent to the NASA Scientific and Technical Information Facility.

If further information concerning this project is desired, please do not hesitate to contact me.

Sincerely,

Thomas J. Mueller
Professor
Aerospace and Mechanical Engineering

cc: Dr. F.M. Kobayashi
Dr. W.B. Berry
Dr. A.A. Szewczyk
✓NASA-STIF

TJM/jk

(NASA-CR-170266) THE STRUCTURE OF SEPARATED FLOW REGIONS OCCURRING NEAR THE LEADING EDGE OF AIRFOILS, INCLUDING TRANSITION Semiannual Status Report, Nov. 1982 - Apr. 1983 (Notre Dame Univ.) 17 p HC A02/MF A01 CSCL 01A G3/02 N83-23274 Unclas 03461

N83-23274#

SEMI-ANNUAL STATUS REPORT

November 1982 - April 1983

for NASA Grant NSG 1419

THE STRUCTURE OF SEPARATED FLOW REGIONS OCCURRING NEAR THE LEADING EDGE OF AIRFOILS - INCLUDING TRANSITION

Thomas J. Mueller
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April 1983

SEMI-ANNUAL STATUS REPORT

November 1982 - April 1983

NSG 1419*

During this report a large amount of data was obtained for the NACA 66_3-018 airfoil in order to assess the advantages (if any) of tripping the boundary layer very near the leading edge for chord Reynolds numbers between 40,000 and 200,000. Single element trips were made from tape 2.5mm wide and 0.15mm thick. The unmodified tape trip in five thickness was placed across the span at 1.1% chord and the results of lift and drag measurements compared with the smooth airfoil case. Saw-tooth geometry trips were also cut from tape and studied using five thicknesses. The saw-tooth trips were placed across the span with the sharp points facing upstream at 1.1% chord and with the valley of the teeth at the 2.5% chord position. A large number of bumps and wiggles were produced in the lift coefficient versus angle of attack curves with various combinations of Reynolds number, thickness, and type of trips. The main result in the drag coefficient was an increase in $C_{d\ min}$ with increase in trip thickness. No overall improvement in airfoil performance could be found for any of the combinations studied. Furthermore, there appeared to be very little difference in measured performance when using either the unmodified tape trip or the saw-tooth tape trip. These results strongly suggest that an airfoil not correctly designed for low Reynolds numbers may not be improved simply by using single element trips.

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*The NASA Technical Officer for this Grant is Mr. Dan M. Somers, NASA Langley Research Center, Hampton, Virginia 23665

The influence of free stream disturbances on the lift and drag performances of the Lissaman 7769 airfoil was also studied in this period. The free stream disturbance level and type affected transition which in turn affected the boundary layer behavior. These experiments covered the chord Reynolds number range from 100,000 to 300,000. The wind tunnel disturbance environment was measured using hot-wire anemometer and sound pressure level equipment. The disturbance level was increased by the addition of a "turbulence screen" upstream of the test section and/or the addition of a flow restrictor downstream of the test section as shown in Figure 1.

The lift and drag performance of the smooth Lissaman airfoil in the standard wind tunnel configuration (i.e., no turbulence screen or flow restrictor–free stream turbulence less than 0.1%) is shown in Figure 2. As the angle of attack was increased, smoke visualization indicated that at 6° the laminar boundary separated on the upper surface at about 25% chord while at 8° the boundary layer appeared to be undergoing transition and separated from the upper surface at about 35% chord. At an angle of attack of 10° transition appeared to be complete and the boundary layer remained attached until about the 70% chord location. There is a noticeable change in the lift curve slope associated with the extension of attached turbulent flow. A smoke photograph at α = 12° is shown in Figure 3a. The lift coefficient continues to increase in this region, Figure 2a, until it reaches a maximum value of 1.3 at 16°. Further increase in angle of attack cause the location of turbulent separation near the trailing edge to move upstream,and C_ℓ to decrease slightly, until it reaches about 35% chord where a jump takes place to a laminar separation at the leading edge at about 19°. At this point there is an abrupt decrease in C_ℓ from about 1.25 to about 0.9. As the angle of attack is decreased from 25°, the boundary layer separates in the laminar state and the C_ℓ remains about 0.9 until an angle of 11° is reached.

With little or no free stream turbulence present the very short laminar boundary layer separates from the airfoil before transition takes place. A comparison of the airfoil flow field at $\alpha = 12°$ for both increasing and decreasing angle of attack is shown in Figure 3. The lift jumps up at $\alpha = 10°$ as a result of the fact that transition in the separated shear layer allows the flow to reattach. The accompanying variation in the profile drag coefficient is shown in Figure 2b. The abrupt decrease in C_ℓ is accompanied by an abrupt increase in C_d. Therefore in the lowest turbulence, quietest tunnel configuration, a significant hysteresis region in the lift and drag forces was found. The presence and extent of this hysteresis was determined by the location of separation and/or transition in the boundary layer. The location of transition from laminar to turbulent flow in the boundary has been known to be affected by the level and type of free stream disturbances for a long time.

In earlier experiments using this airfoil hysteresis was not found. These data were taken by increasing the angle of attack from -10° to +20° angle of attack and then turning the tunnel off for the balance calibration. The airfoil was then returned to -10° angle of attack for the next experiment. In the present investigation no attempt was made to determine whether or not hysteresis occurred at negative angles of attack.

The result of changing the acoustical environment by adding one flow restrictor at the end of the test section is shown in Figure 4. The addition of one restrictor increases both the free stream turbulence level and the sound pressure level since the fan speed must be increased for a fixed value of tunnel velocity. This test section environment reduced the size of the hysteresis region and produced a slightly higher $C_{\ell max}$ of almost 1.4. A slightly lower minimum drag coefficient was also obtained. The use of two

flow restrictors (i.e., still higher fan speed) produced similar results with the hysteresis being almost completely eliminated. The increase in free stream turbulence and acoustic excitation caused the laminar shear layer to transition much earlier, thus allowing the flow to reattach sooner.

Increasing the free stream turbulence level to about 0.3% , by adding one 7.09 meshes/cm screen at the upstream end of the test section with no flow restrictor, produced the lift and drag coefficients presented in Figure 5. This test section environment completely eliminated the hysteresis region and yielded values of $C_{\ell max}$ and $C_{d\ min}$ between those of Figures 2 and 4. With a larger turbulence intensity in the test section, the airfoil boundary layer transitions very close to the leading edge, eliminating hysteresis by enabling the flow to reattach at higher angles of attack. The abrupt decrease in C_ℓ occurred at approximately the same angle of attack in each case. The very large adverse pressure gradient at this angle of attack (i.e. 19°) caused the boundary layer to separate whether it was laminar or turbulent. Hysteresis occurred because the laminar separated shear layer did not reattach. An increase in turbulence did not prevent the abrupt loss of lift, but the separated flow was turbulent allowing more rapid reattachment.

When the chord Reynolds number was increased to 200,000 the hysteresis region was reduced when using the standard wind tunnel configuration. At this condition the abrupt decrease in lift occurred at about 19° for increasing angle of attack and the lift jumped up when the angle of attack was decreased to 16°. At a chord Reynolds number of 300,000 the abrupt decrease in lift occurred at about 21° and jumped back up at about 20°.

The importance of this hysteresis phenomena cannot be overemphasized. Low Reynolds number airfoil data obtained in noisy and/or high turbulence wind tunnels may not exhibit significant hysteresis. Therefore, aircraft designed using such wind tunnel data may not perform as expected in flight where the

free stream disturbance level is usually very low.

Free-stream disturbances are a major source of disparity in experimental data. However, there are other sources of disparity which produce results similar to those produced by free-stream turbulence. Figures 6a and 6b show the lift and drag curves produced in the standard wind tunnel environment with a strip of tape 2.5mm wide and 0.15mm thick placed near the leading edge (i.e. across the span at 1.1% chord) of the airfoil. This small boundary layer trip reduced the hysteresis in a similar manner to the introduction of a flow restrictor. The tape produces similar results by tripping the boundary layer and causing early transition. A model with a small amount of surface roughness or irregularities in the surface caused by fabrication defects could produce the same results.

The problems associated with obtaining accurate wind tunnel data for airfoil sections at low Reynolds numbers are compounded by the extreme sensitivity of the boundary layers to the free stream disturbance environment. The effect of free stream disturbances varies with magnitude, frequency content, and source of the disturbance. The sensitivity and accuracy of the measurement and data acquisition systems as well as the experimental procedure used can have a substantial effect on the results obtained.

Although free-stream disturbances produced the largest disparity between different tests for the Lissaman airfoil, not all of the differences can be attributed to free-stream disturbances. Model imperfections or surface roughness can produce results identical to those achieved with free stream disturbances. Reynolds number effects are critical at low speeds. An increase in Reynolds number from 150,000 to 200,000 will eliminate a major portion of the hysteresis, and the hysteresis is insignificant at 300,000. It is important that the free-stream disturbances be well documented for each

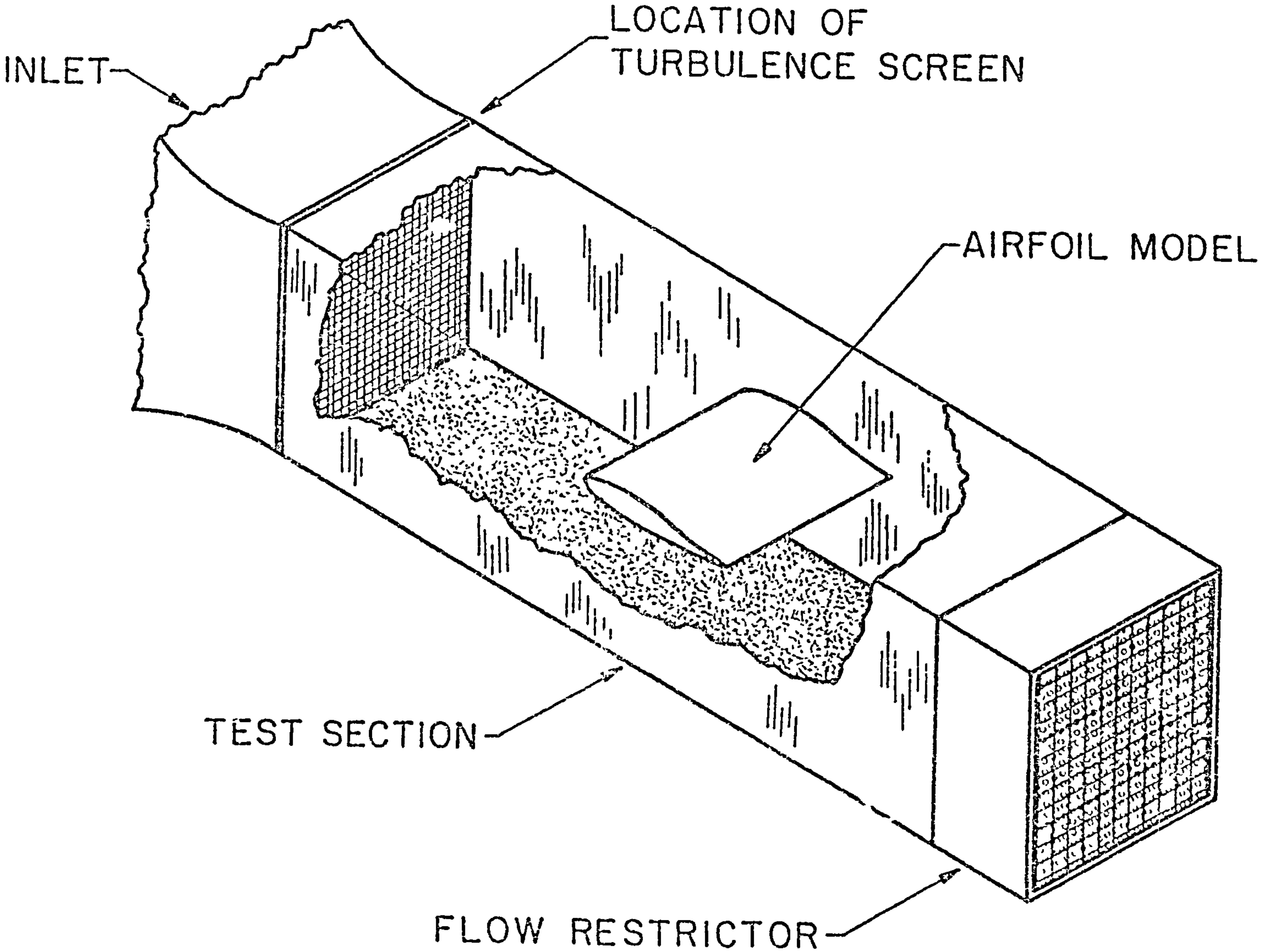
test condition in order to correctly attribute differences in test results to these free stream disturbances. A clear distinction between the effects of free-stream disturbances, model irregularities, and Reynolds number must be made before the performance of airfoils at these Reynolds numbers can be clearly understood.

This investigation indicates that it should not be surprising that different low Reynolds number results are obtained from different wind tunnel laboratories.

During this report period the following papers produced under this grant were published:

1. "An Experimental Investigation of the Low Reynolds Number Performance of the Lissaman 7769 Airfoil," P.E. Conigliaro, AIAA Paper No. 83-0647.

2. "The Influence of Free-Stream Disturbances on Low Reynolds Number Airfoil Experiments," T.J. Mueller, L.J. Pohlen, P.E. Conigliaro, and B.J. Jansen, Jr., in Experiments in Fluids, Vol. 1, pp. 3-14, 1983.


LOCATION OF
TURBULENCE SCREEN
INLET
AIRFOIL MODEL
TEST SECTION
FLOW RESTRICTOR

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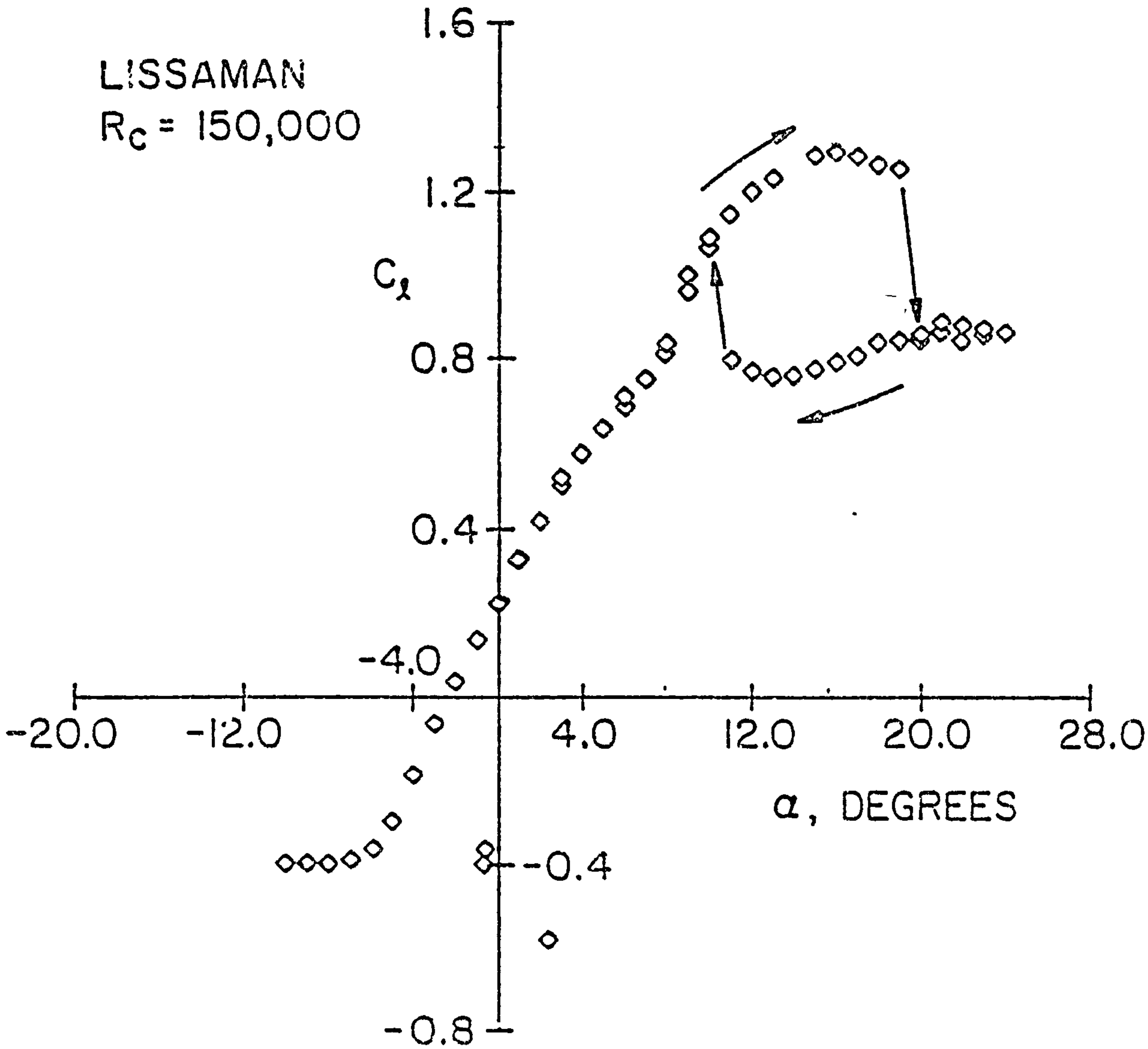
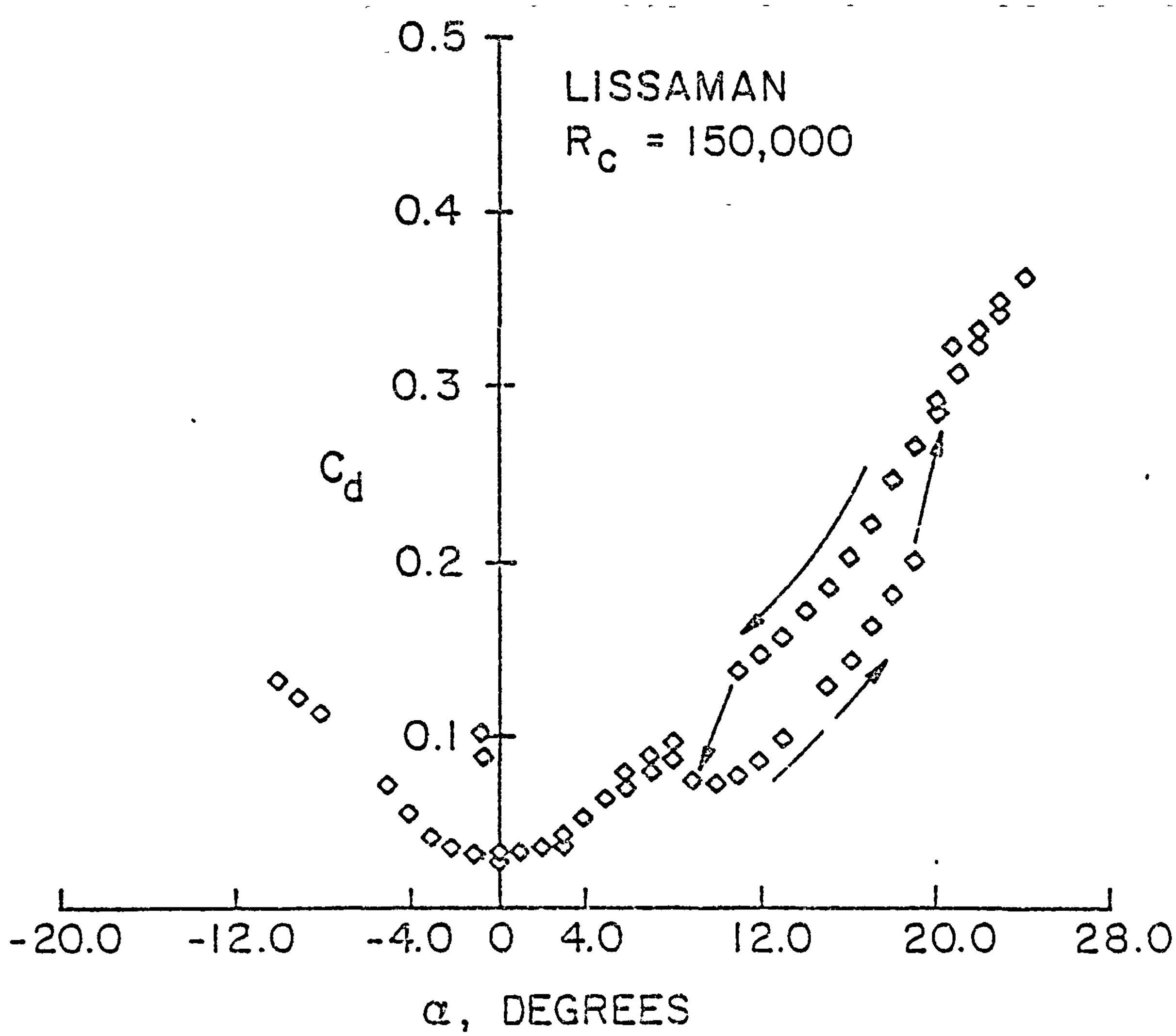


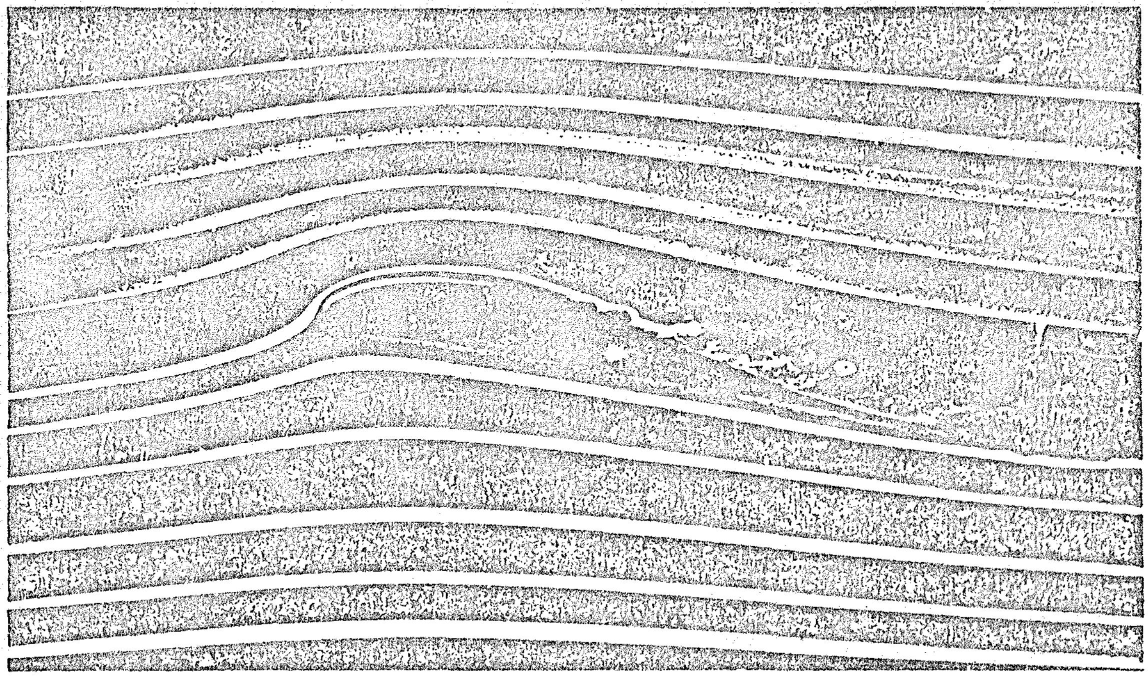
a.) Section Lift Coefficient

Figure 2. Lift and Drag Coefficients versus Angle of Attack of the Smooth Lissaman Airfoil with No Screen or Flow Restrictor (Hysteresis).

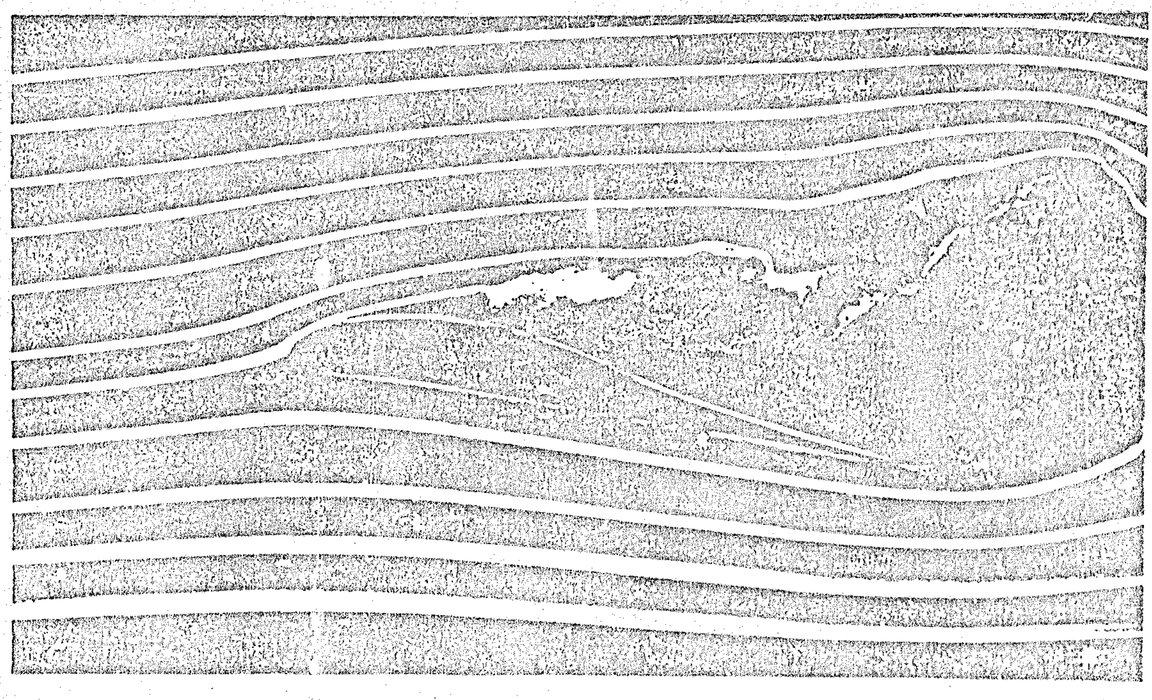


b.) Section Profile Drag Coefficient

Figure 2. Lift and Drag Coefficients Versus Angle of Attack of the Smooth Lissaman Airfoil with No Screen or Flow Restrictor (Hysteresis).

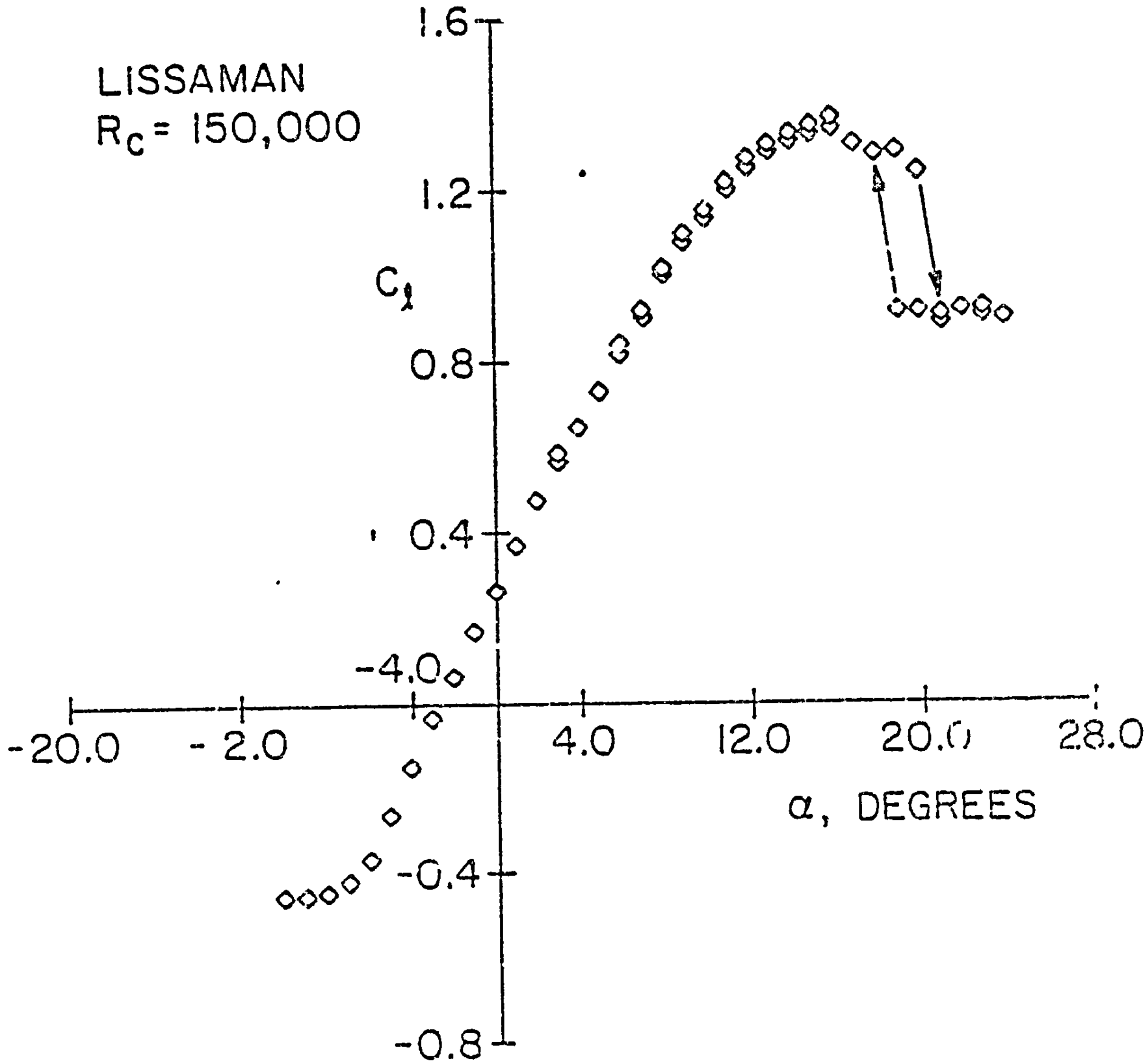


a.) Increasing Angle of Attack



b.) Decreasing Angle of Attack

Figure 3. Smoke Photographs of Lissaman Airfoil at R_c = 150,000
at 12° Angle of Attack With No Flow Restrictor or Screen.



a.) Section Lift Coefficient

Figure 4 . Lift and Drag Coefficients Versus Angle of Attack of the Smooth Lissaman Airfoil With No Screen and One Flow Restrictor.

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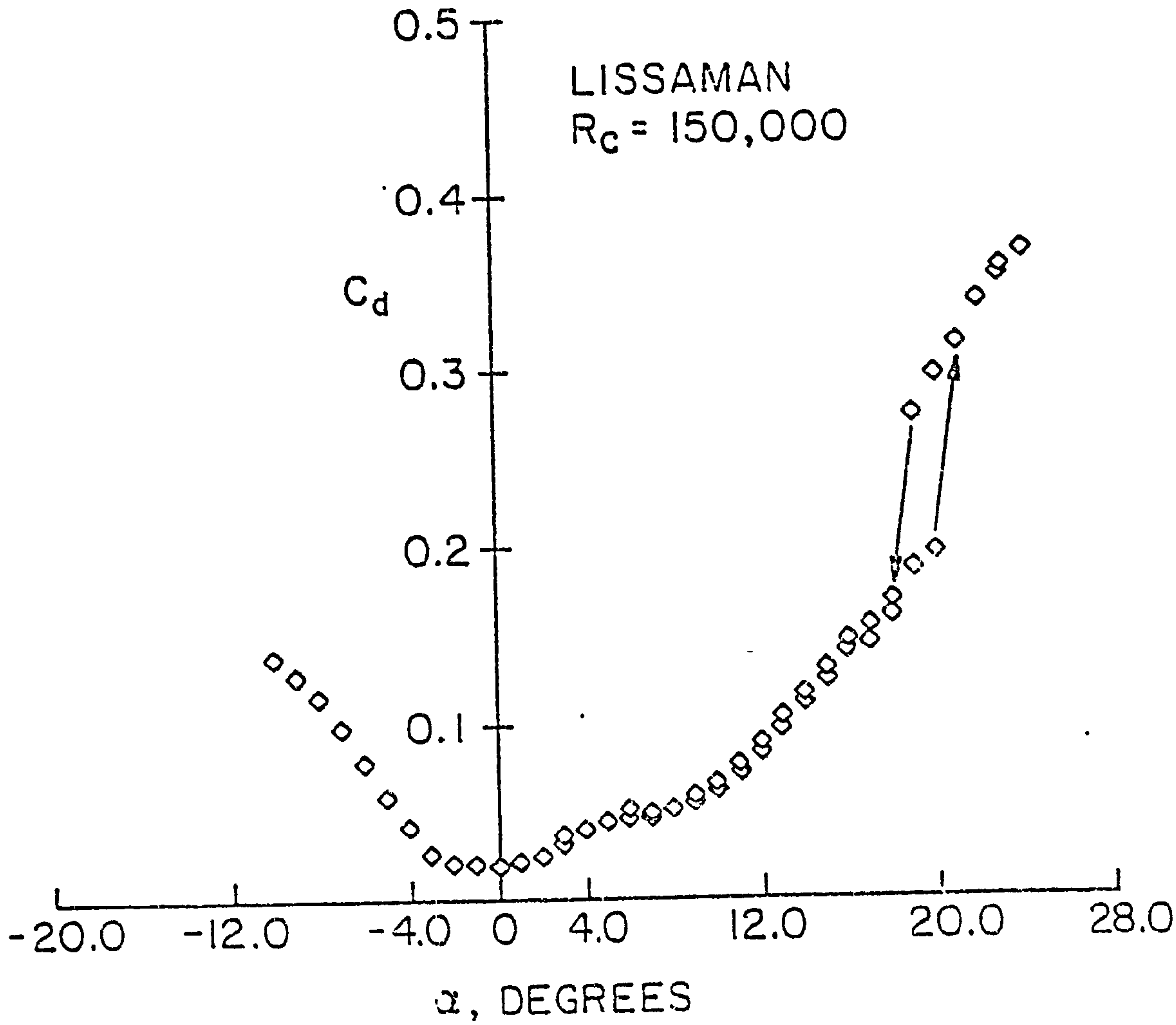
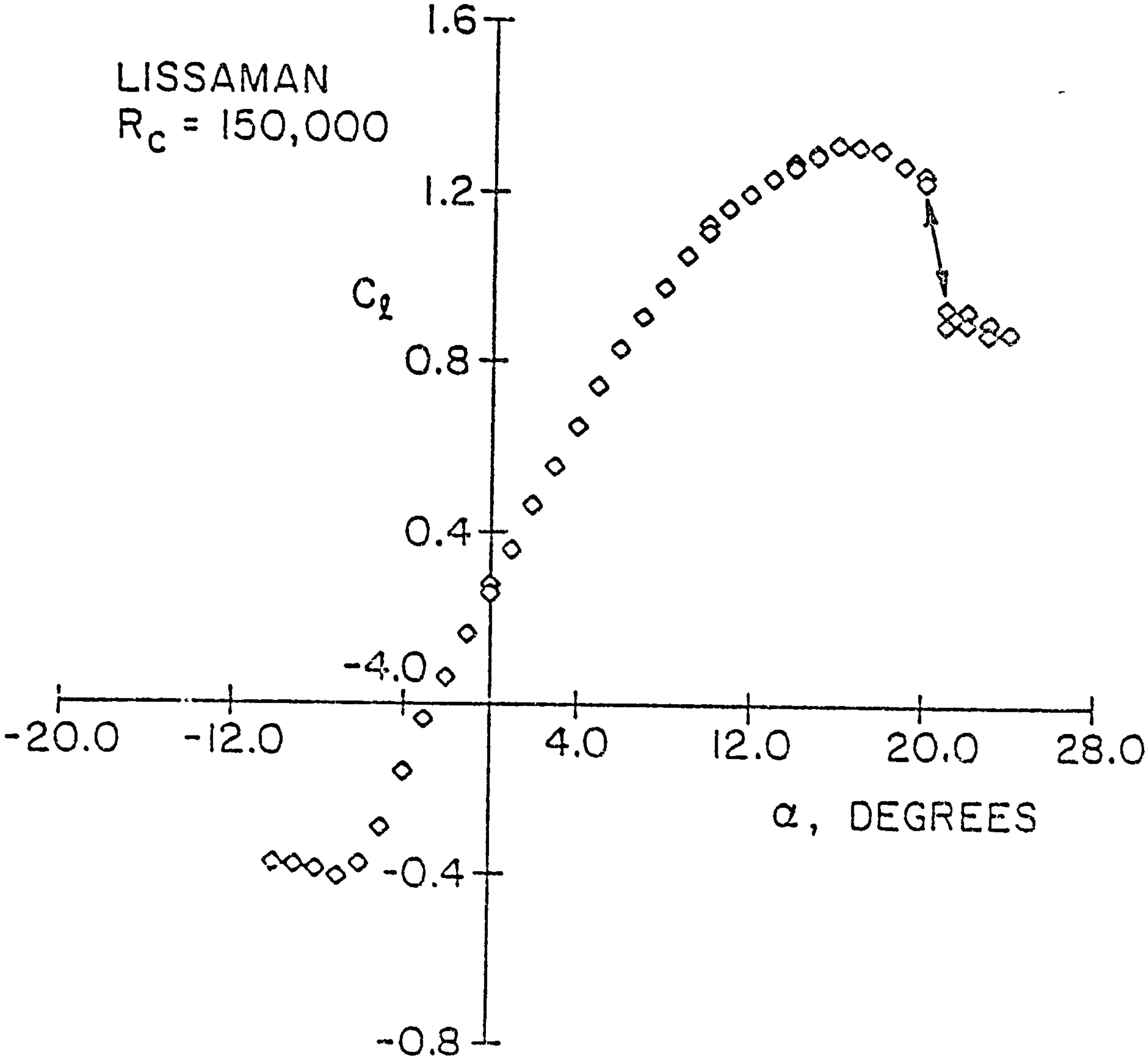


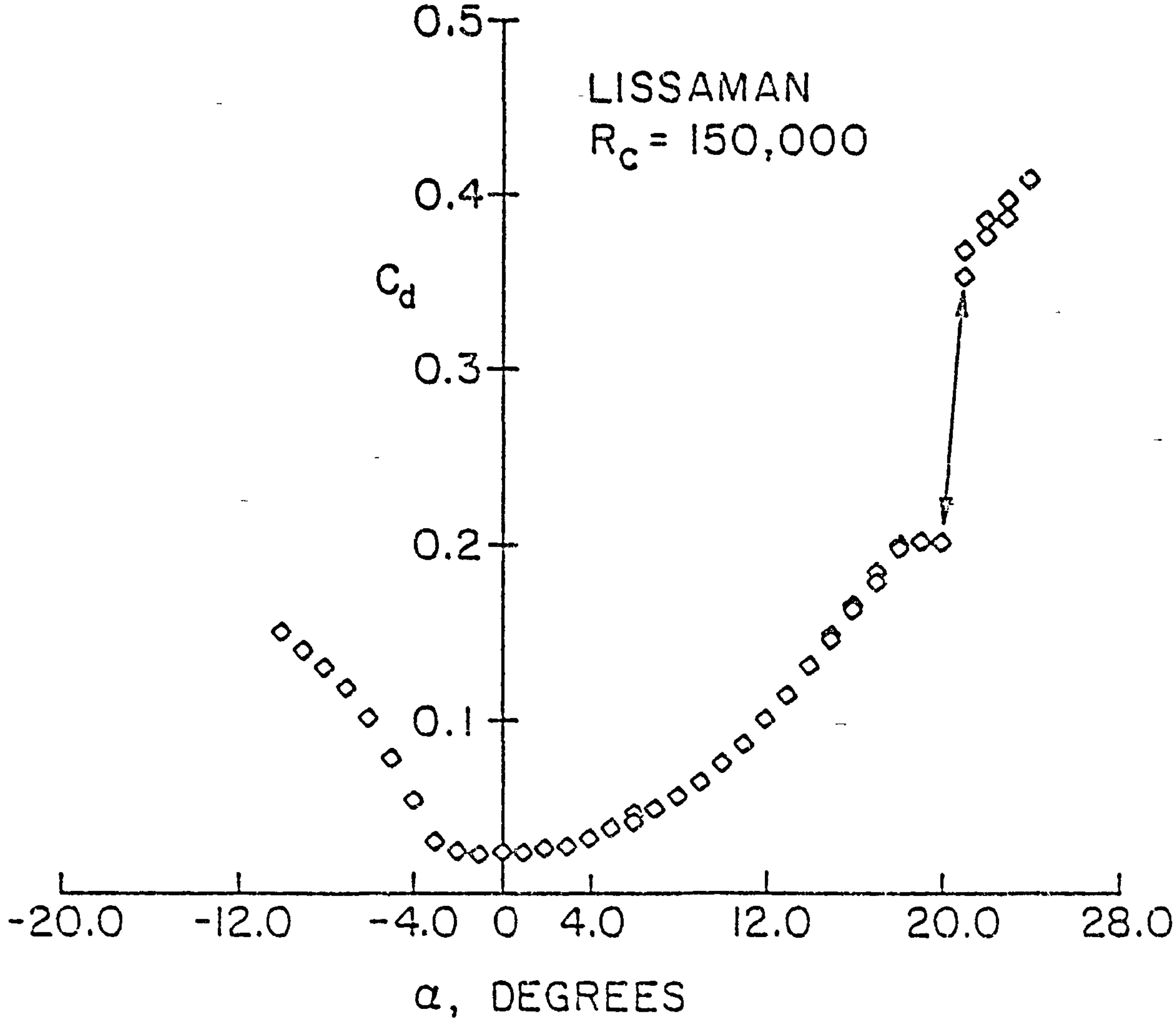
b.) Section Profile Drag Coefficient

Figure 4. Lift and Drag Coefficients versus Angle of Attack of the Smooth Lissaman Airfoil With No Screen and One Flow Restrictor.



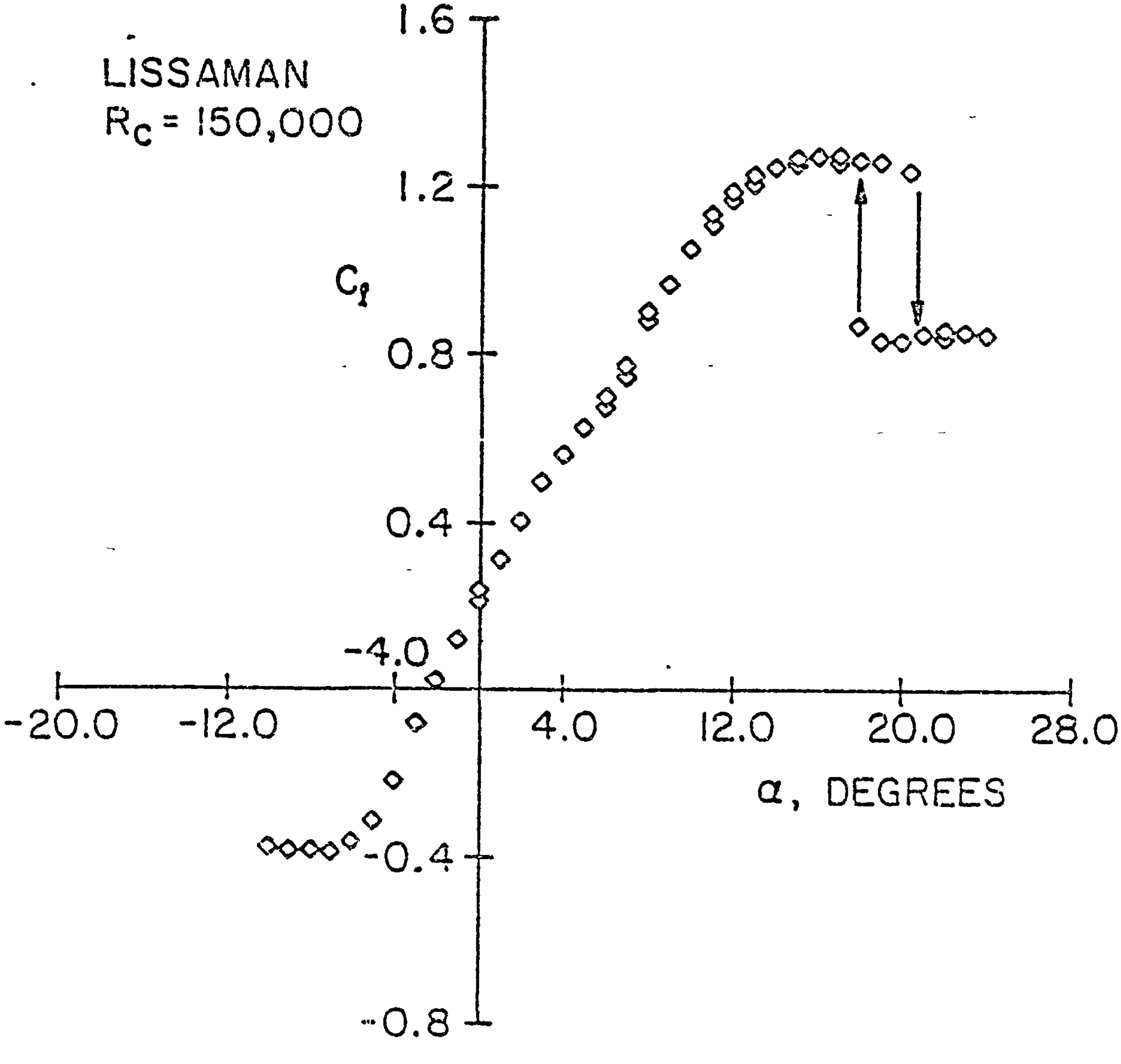
a.) Section Lift Coefficient

Figure 5 Lift and Drag Coefficients versus Angle of Attack of the Smooth Lissaman Airfoil with One 7.09 Meshes/cm Screen and No Flow Restrictor.



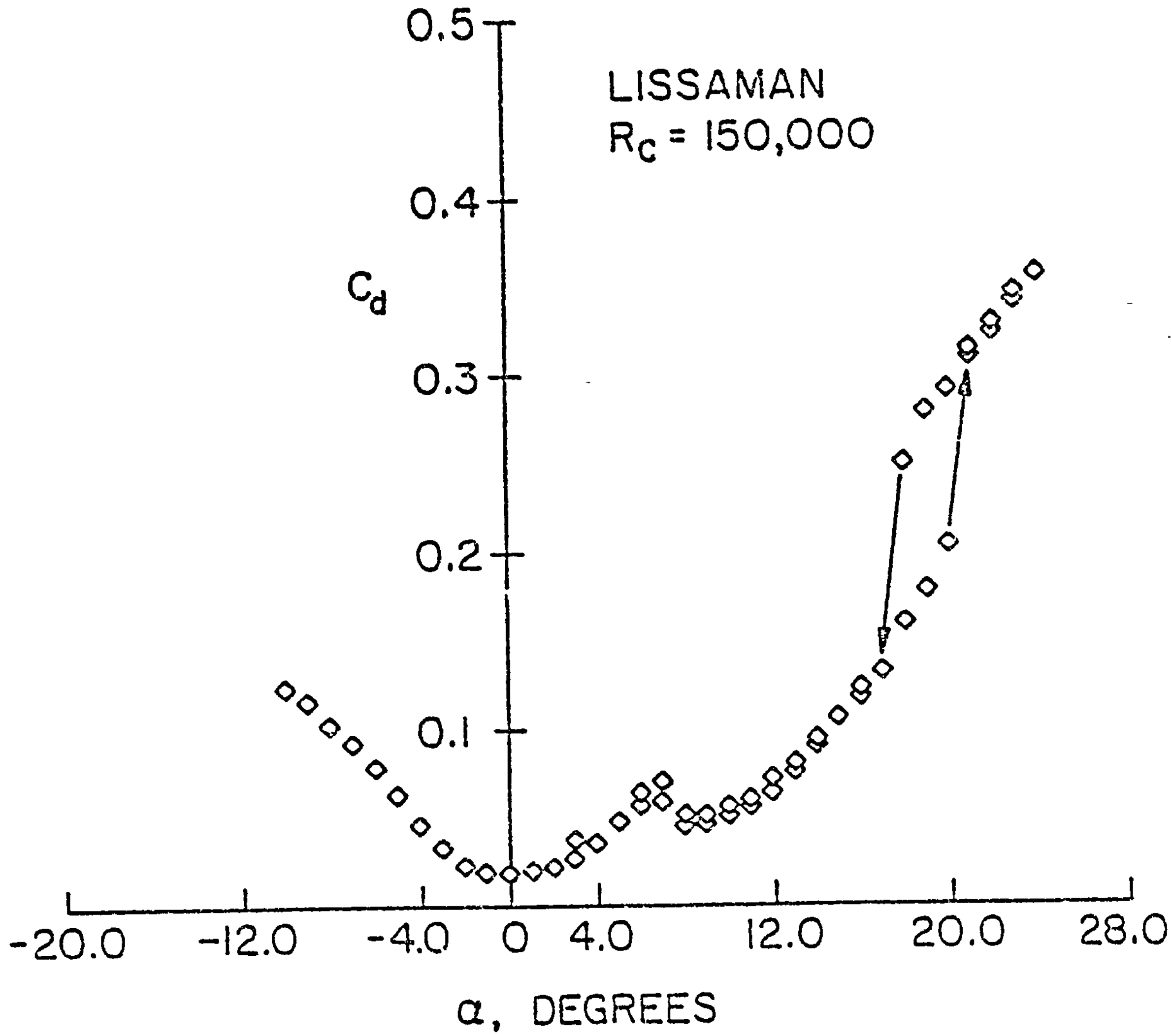
b.) Section Profile Drag Coefficient

Figure 5. Lift and Drag Coefficients versus Angle of Attack of the Smooth Lissaman Airfoil with One 7.09 Meshes/cm Screen and No Flow Restrictor.



a.) Section Lift Coefficient

Figure 6. Lift and Drag Coefficients versus Angle of Attack of the Lissaman Airfoil with Tape Trip at 1.1% Chord and No Screen or Flow Restrictor.



b.) Section Profile Drag Coefficient

Figure 6. Lift and Drag Coefficients versus Angle of Attack of the Lissaman Airfoil with Tape Trip at 1.1% Chord and No Screen or Flow Restrictor.

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